

NEGATIVE C3 LAUNCH OPTIONS FOR SOLAR SYSTEM EXPLORATION

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Abstract

Low thrust trajectory analyses were used to examine the feasibility of using solar electric propulsion for Earth escape from a negative C3 launch for deep space missions in order to significantly increase the net delivered mass capability of inexpensive launch vehicles. Derivatives of Hall thruster systems that are likely to be developed for commercial applications were selected to provide the Earth escape propulsion in the expectation that such systems will become widely available and inexpensive, and because these systems are projected to have performance (Isp and thrust-to-power) characteristics that are attractive for this application. Ion engines like those planned for use on NASA's Deep Space IV mission and high specific impulse Hall thrusters were considered for the heliocentric mission phase. The Hall thruster power assumed for these analyses ranged from 3.4 kW to 5 kW. Different cases were examined for solar array power at beginning of life of about 18 kW and 23 kW. Two example missions were examined: a Europa orbiter mission and a Europa lander mission. The results of this study show that use of solar electric propulsion is mission-enhancing for a Europa orbiter mission; it may enable the use of a lower cost launch vehicle than can be used for all-chemical propulsion options. Solar electric propulsion in combination with negative C3 launches may be mission enabling for a Europa lander mission if high performance Hall thrusters (5 kW, 3000 s Isp) with the necessary lifetime become available. The geocentric portion of this analysis may be used as a tool for examining potential benefits for other deep space missions.

Introduction

In a time when use of Solar Electric Propulsion (SEP) systems on commercial spacecraft is becoming common, and with the possibility that commercial

applications of electric thrusters for orbit transfer and station keeping may force a significant reduction in the cost of these systems, there is an enormous potential for the use of such systems as low-cost modules for interplanetary spacecraft deployed from Earth orbit (i.e. with negative hyperbolic excess energy, or C3). Negative C3 launches coupled with the use of SEP to spiral out of Earth orbit could significantly increase the net payload mass delivered to various planetary objectives relative to traditional chemical propulsion systems. Furthermore, negative C3 launches permit the use of moderately priced launch vehicles to achieve ambitious payload mass objectives, as shown in Figure 1.

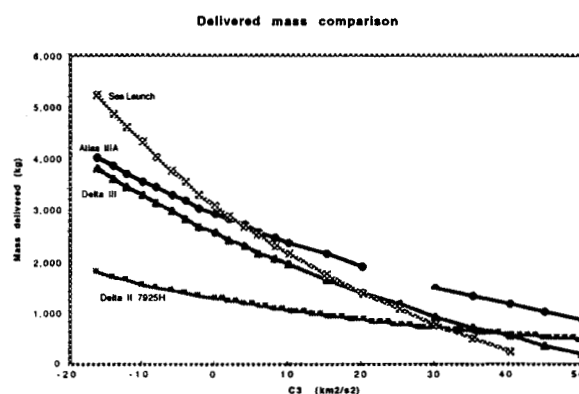


Figure 1. Delivered mass as a function of C3 for the Delta II 7927, Delta III, Atlas IIIA, and Sea Launch.

The advantages of significantly increasing net spacecraft mass must be weighed against the added trip time required for the Earth-spiral portions of such missions. In addition, the cost and complexity of high power solar arrays and the effects of radiation from the Earth's radiation belts must be evaluated. Added mission complexity and operational costs are also concerns.

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Finally, possible impacts on science objectives due to the large solar array size must be considered.

Several factors may now make the use of negative C3 launches more attractive for deep-space missions. The first of these is the requirement for deep-space missions to account for their entire life-cycle costs. The use of SEP coupled with a negative C3 launch may enable the use of a smaller, less expensive launch vehicle. The launch vehicle savings, however, will be offset by the additional cost of the SEP system, including the solar array, and any additional operations costs due to longer flight times. This approach, therefore, will benefit strongly from the commercial development of low-cost Hall-thruster propulsion systems.

Several types of electric propulsion systems are currently in use on commercial satellites, including hydrazine resistojets on the Iridium constellation of satellites, xenon ion engines on two Hughes-built HS-601HP satellites, and xenon Hall thrusters on the Russian GALS and EXPRESS series of satellites. It is anticipated that many of the proposed large LEO satellite constellations, such as Teledesic, Celestri, Iridium-Next, and SkyBridge will benefit from the use of electric propulsion. Furthermore, these constellations may consist of hundreds of satellites; the use of electric propulsion on this scale will result in significantly lower hardware costs. Because the performance characteristics of Hall thrusters make them attractive candidates for these satellite constellations, it is assumed in this study that a low-cost Hall-thruster-based propulsion system will be selected and developed for one or more of them.

Finally, this study included evaluation of Delta-class (7290, 7925H, and Delta III), Atlas IIIA, and Sea Launch launch vehicles.

In general, when comparing chemical (~ 300 s Isp) and SEP systems (~ 2000 s Isp) for Earth escape, there will only be a benefit to using a SEP module (including solar array mass) and a negative C3 launch if the SEP system dry mass is less than about 27% of the launch mass (or 39% of the payload mass delivered to a C3=4 km^2/s^2). This rule of thumb is conservative; in using it, one assumes no part of the SEP module used for Earth escape can be used during a deep space mission phase. In this study, two examples that meet the 27% maximum SEP mass requirement are examined.

The two examples are both missions to Jupiter's moon Europa. One mission scenario is for a Europa orbiter

spacecraft to be launched in the 2002 to 2004 time frame. The other is the launch of a Europa lander spacecraft in the 2006 to 2009 time frame. These examples may be particularly well suited applications of negative C3 launches, as the hostile radiation environment at Europa (or other Jovian moons) will require a spacecraft design that will easily be tolerant of the much lower particle fluence in the Earth's radiation belts.

The purpose of this study is to examine potential benefits to deep space missions in terms of lower launch costs and higher payload mass delivery to propulsively difficult deep-space destinations using solar electric propulsion and negative C3 launches.

In the following sections, the methodology used to evaluate this approach, along with detailed propulsion subsystem mass lists, solar array performance data, radiation fluence data, launch vehicle performance, and SEP system performance (for delivery to a positive C3 of 4 km^2/s^2), are presented. Finally, the Europa orbiter and lander mission examples are discussed.

Methodology

The mission trajectory analyses used in the evaluation methodology are performed in two parts. The first determines the mass and trip time required to transfer the spacecraft from the initial Earth orbit to an Earth escape C3 of 4 km^2/s^2 as a function of initial C3 for each launch vehicle. A starting perigee of 500 km is assumed in all cases to minimize concerns regarding the effects of atmospheric drag. The use of 3.4 kW Hall thrusters operating at 1800 s specific impulse, 200 mN, and 52% efficiency is assumed for the SEP system. This segment is referred to as the geocentric phase.

The second segment, or heliocentric phase, is an optimization of mass delivered to a target as a function of initial mass for heliocentric mission trajectories starting from a C3 of 4 km^2/s^2 . The trajectory is optimized to maximize the final mass for a given beginning-of-life solar array power referenced to 1 AU. This optimized final payload mass is then compared with the mass needed to meet the objectives of the mission examples chosen for this study. If the optimized final payload mass is greater than or equal to the required mass, then the corresponding initial mass is compared with that delivered to Earth escape as calculated in the geocentric phase analysis.

Electric propulsion systems employing ion thrusters as required for NASA's Deep Space IV mission and high specific impulse Hall thrusters are considered for the

heliocentric mission phase in terms of the mass assumed for each system. All trajectories were calculated using NSTAR throttling profiles. The number of thrusters were varied to use all available solar array power.

The heliocentric phase is accomplished with SEP and gravity assists. In the case of the examples studied here, the SEP system is discarded prior to Jupiter Orbit Insertion (JOI), which is done with chemical propulsion. The desire for short trip times dictates high arrival speeds at Jupiter, and consequently the need for a large chemical propulsion system for JOI. The remainder of the mission is performed chemically. The amount of chemical propellant required is dependent upon the hyperbolic excess speed at arrival (V_{hp}), and various each SEP trajectory. Roughly, each additional 1 km/s arrival speed will result in the need for approximately 100 m/s more ΔV for capture.

Study Parameters

With one exception, the technology level for propulsion and power subsystems assumed for this study is state-of-the-art. The thruster parameters used for the trajectory calculations are consistent with performance demonstrated in ground-based endurance tests. For example, Table 1 shows performance data for the Primex SPT-4000 Hall thruster [3].

Discharge Power (W)	Volts	Amps	Thrust (mN)	Specific Impulse (s)	Vacuum (Torr. Xe)
600	150	4.0	36	783	1E-5
600	350	1.7	23	924	4E-6
1700	200	8.5	119	1296	4E-5
1700	350	4.9	99	1709	2E-5
3400	300	11.3	212	1786	5E-5
3400	350	9.7	198	1933	4E-5
3400	350	9.7	205	1947	4E-5
3400	400	8.5	188	2035	4E-5
4000	350	11.4	240	1974	5E-5
4000	350	11.4	246	2039	5E-5
4000	350	11.4	248	2056	5E-5
4000	350	11.4	245	2031	5E-5
4500	350	12.9	271	2008	6E-5
4500	400	11.3	255	2148	5E-5

Table 1. Primex BPT-4000 performance data. Engine life at 4.0 kW projected to be in excess of 6000 hours. Thrust accuracy $\pm 2\%$, specific impulse accuracy $\pm 3\%$. [3].

The development of high Isp Hall thrusters is the least mature technology assumed here. The assumption of sufficient thruster lifetime at a specific impulse of approximately 3000 s is based on the calculations by Clauss et al.[ref]. Clauss and his coauthors show an estimate of reduction in total impulse attributed to decreased thruster life at high discharge voltage (36% reduction from 300V to 700V). A discharge voltage of 700 V corresponded to a specific impulse of 3100 s in the SPT-200. Throughout this study, a total propellant throughput maximum of 325 kg of xenon is assumed, regardless of the operating power.

Electric Propulsion System

During the geocentric mission phase, the Hall thrusters are assumed to operate with an input power of 3.4 kWe (3.7 kWe input power to the Power Processing Unit PPU), a specific impulse of 1800 s, a thrust of 200 mN, and an efficiency of 52 percent. These performance parameters are held constant until Earth escape. At that time, the Hall thruster module (including propellant tank) is jettisoned, and a second SEP module based on either higher Isp Hall thrusters or NSTAR-derivative ion engines, is used for the heliocentric phase.

The NSTAR-derivative ion thrusters are assumed to have a beginning-of-life operating specific impulse of 3100 s. Total ion engine propellant throughput is assumed to be consistent with DS-4 mission requirements, which is approximately 50% greater than those of DS-1 - the first demonstration mission of NSTAR technology scheduled for launch this fall. The system mass for such a module is assumed to be the same as in the DS-1 NSTAR system. This rather conservative estimate provides an upper bound to the heliocentric SEP system mass. However, invoking high performance Hall thrusters with a throttling profile like the ion engine (i.e. without modifying the trajectory calculations) allows the estimate of a high power Hall thruster module mass that is substantially lower.

The propulsion system mass list used in this study is shown in Table 2. It is compared with component masses for DS-4, DS-1, and those assumed in a previous study[1]. Table 3 shows the entire SEP system mass for the geocentric mission phase at two different power level, while Table 4 shows the heliocentric SEP system mass for the examples of a 15 kW Europa orbiter mission using an indirect trajectory (no gravity assists), and a 20 kW Europa lander mission using a single Venus gravity assist.

<i>Mass per unit (kg)</i>	DS-4 Champollion 12-kW	NSTAR (New Millenium DS-1) 2.5 kW	AIAA 97-2782 J. R. Brophy 8.8 kW	This Study 11.1 - 18.5 kW
<i>Hall system</i>				
Engines			2.5 (2.3-kW)	5 (3.4-kW)
PPUs			1.3 (Direct drive)	6.3 * Pppu (kW)
Gimbals			0.8	1.25
PPU thermal control			0.5	20 * Pppu radiated (kW)
DCIU			1	(incl. in NSTAR DCIU)
Fixed feed system			1.6	10
Feed system per engine			1.2	1.5
<i>NSTAR system</i>				
Engines	8.4	8.3	8.2	8.3
PPUs	9.8	14.9	11.9	14.9
Gimbals	7.2	18.8	2.5	7.2
PPU thermal control		3.5	3.5	3.5
DCIU	0.64	2.5	1.9	3
Fixed feed system	7.2	20.5	6.5	10
Feed system per engine	1.2	-	1.2	1.5
Propellant tankage factor	9 %	9 %	10 %	10 %

Table 2. System component mass comparison

Item	QTY	Mass/unit (kg)	Total mass (kg)
Hall stage			
Engines (3.4-kW)		5	
PPUs		= 6.3 x Pppu (kW)	
Gimbals		1.25	
PPU thermal control		= 20 x Pradiated (kW)	
Propellant tank		10 % geocentric prop. mass	
Ion stage			
Engines (2.3-kW)		8.3	
PPUs		14.9	
Gimbals		7.2	
PPU thermal control		3.5	
Propellant tank		10 % heliocentric prop. mass	
DCIU		3	
Gimbal drive electronics		0.5	
Fixed feed system		10	
Feed system per engine		1.5	
PMAD specific to SEP		5	
Structure/cablg per engine		5	
Subtotal 1			
Contingency (30 %)			
Cabling		10% of subtotal + contg.	
Structure/Mechanisms		7.5% of subtotal + contg.	
Thermal		5% of subtotal + contg.	
Separation mechanisms			
Subtotal 2			
Solar arrays			
S/A support structure		10% of solar arrays	
S/A drive		10% of solar arrays	
System Dry Mass			
Total Propellant Mass			
Residuals		2.5% of prop. mass	
System Total wet Mass			

Table 3. System masses of the Hall thruster module components for the geocentric phase and Ion engine system components for the heliocentric phase.

Item	QTY	Mass/unit (kg)	Total mass (kg)
Hall 1rst stage			
Engines (3.4-kW)		5	
PPUs		= 6.3 x Pppu (kW)	
Gimbals		1.25	
PPU thermal control		= 20 x Pradiated (kW)	
Propellant tank		10 % geocentric prop. mass	
Hall 2nd stage			
Engines (3.4-kW)		5	
PPUs		= 6.3 x Pppu (kW)	
Gimbals		1.25	
PPU thermal control		= 20 x Pradiated (kW)	
Propellant tank		10 % heliocentric prop. mass	
DCIU		3	
Gimbal drive electronics		0.5	
Fixed feed system		10	
Feed system per engine		1.5	
PMAD specific to SEP		5	
Structure/cablg per engine		5	
Subtotal 1			
Contingency (30%)			
Cabling		10% of subtotal + contg.	
Structure/Mechanisms		7.5% of subtotal + contg.	
Thermal		5% of subtotal + contg.	
Separation mechanisms			
Subtotal 2			
Solar arrays			
S/A support structure		10% of solar arrays	
S/A drive		10% of solar arrays	
System Dry Mass			
Total Propellant Mass			
Residuals		2.5% of prop. mass	
System Total wet Mass			

Table 4. System masses of the Hall thruster module components for the geocentric phase and high performance Hall thruster system components for the heliocentric phase.

Solar Arrays

A survey of present and near term solar array technology was made and is summarized in Tables 5 and 6. Solar arrays were sized for the example missions by determining the array area required at the end of the

geocentric mission phase to deliver enough power for operation of the thrusters at the design point of 3.7 kWe input to the PPU per thruster (14.8 kWe and 18.5 kWe array for 4 and 5 thrusters, respectively).

Cell type	Efficiency (1 AM0 28°C)	Efficiency @ 1AU (Temperature factor included)	Default Temp. Coefficient (> 28°C)
Si K4710 Spectrolab	14 %		- 0.435
Hi-Eff Si 150m Sun Power	18 %	16.8 %	
Hi-Eff Si SHARP	18 %	16.8 %	
GaAs/Ge 175m Spectrolab	19 %	18.4 %	
GaAs/Ge 25th IEEE PVSC	19 %	17.8 %	
GaAs/Ge 135m TechStar	19 %	17.7 %	
GaInP2/GaAs DJ 140m Spectrolab	21.5 %	20.7 %	
GaInP2/GaAs DJ Spectrolab 1994 IEEEWC	21.5		
GaInP2/GaAs DJ 135m TechStar	21.5	20.8 %	
CIS - Advanced	11 %		- 0.46
GaInP2/GaAs/Ge TJ 140m Spectrolab	24 %		
GaInP2/GaAs/Ge TJ Spectrolab 1994 IEEEWC	24 %		
GaInP2/GaAs/Ge TJ Spectrolab 1996 IEEEPVSC	24 %		

Table 5. Types and performances of solar array cells.

Array type - cell type	Coverglass (mils)	W/kg (BOL or otherwise specified)	W/m ² (BOL)	kg/m ²	Comments
Generic rigid/planar array - Si	3	55			Derived from Mars Pathfinder type
Generic rigid/planar array - Hi Eff Si	3	86	199	2.31	Mars Pathfinder type
Generic rigid/planar array - GaAs	3	68	199	2.93	Derived from Mars Pathfinder type
Generic rigid/planar array - DJ	3	77	225	2.92	Derived from Mars Pathfinder type
Generic rigid/planar array - TJ	3	87	255	2.93	Derived from Mars Pathfinder type
PCI array - Hi Eff Si	4	81	73	0.9	
AFPL array - Hi Eff Si	4	62	104	1.67	
AEC ABLE Puma array	3	47			
AEC ABLE SCARLET array - GaAs	6	60			
AEC ABLE SCARLET array - DJ	6	70			
TRW APSA array - Si+GaAs	2	80			
L'GARDE Inflatable array - CIS	3	96	93	0.97	
L'GARDE Inflatable array - C-Si	3	100 (EOL)			

Table 6. Solar array technologies.

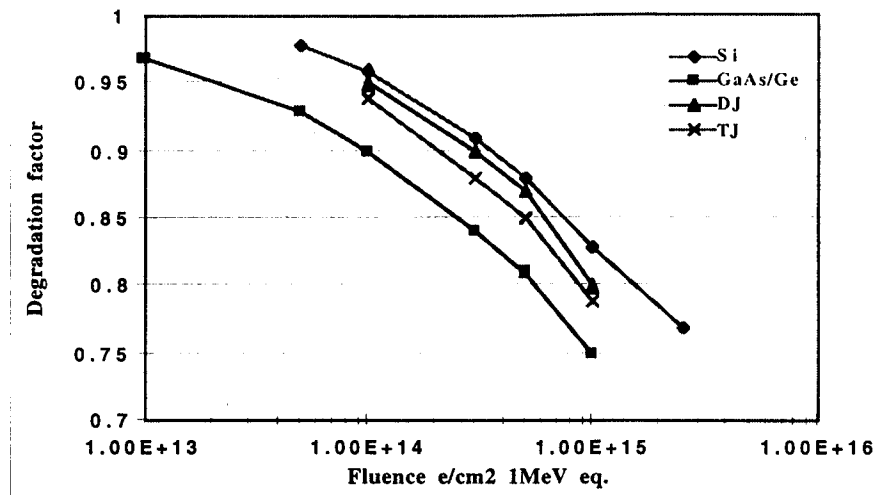


Figure 2. Degradation factor for solar cells.

Cell efficiency @ 1 AU	See Table 6
<i>Packaging and fabrication losses</i>	
Mismatch & fabrication	0.98
Wiring & diode loss	0.96
Packing factor	0.85
<i>Environment losses</i>	
Temperature loss factor	included
Shadowing losses	1.00
Sun offset angle	0
Solar cosine loss	1.00
<i>Life factors</i>	
UV	0.98
Radiation	see table/fig x
Fatigue (thermal cycling)	0.98
Micrometeoroids	0.98

Table 7. Solar array sizing assumptions. Note that 6 mils of CMX coverglass at 2.25 g/cm³ is assumed.

Radiation

Calculation of the radiation fluence in 1 MeV electron equivalents was made for five different spiral orbit scenarios corresponding to different starting altitudes and trip times. These five cases are indicated by the triangular points in Figure 2, and form an envelope in trip time and starting altitude representative of the other

scenarios examined. Solar maximum conditions were assumed. The 1 MeV electron equivalent fluence is indicated by the shaded squares.

The trajectories used to calculate the charged particle fluences indicated in Figure 2 were based on SEP thruster performance of 2000 s specific impulse and 270 mN thrust. However, the trajectories were initiated in circular parking orbits, not the elliptical starting orbits shown in Figure 3, thereby giving longer trip times for the same launch vehicle. Hence, the radiation calculations used here represent a worst-case scenario. Only trapped radiation was considered; radiation exposure occurring during the heliocentric mission phases (due to solar flares) is not considered. Nevertheless, solar flare damage will in all likelihood be significantly less than that acquired during the Earth spiral-out.

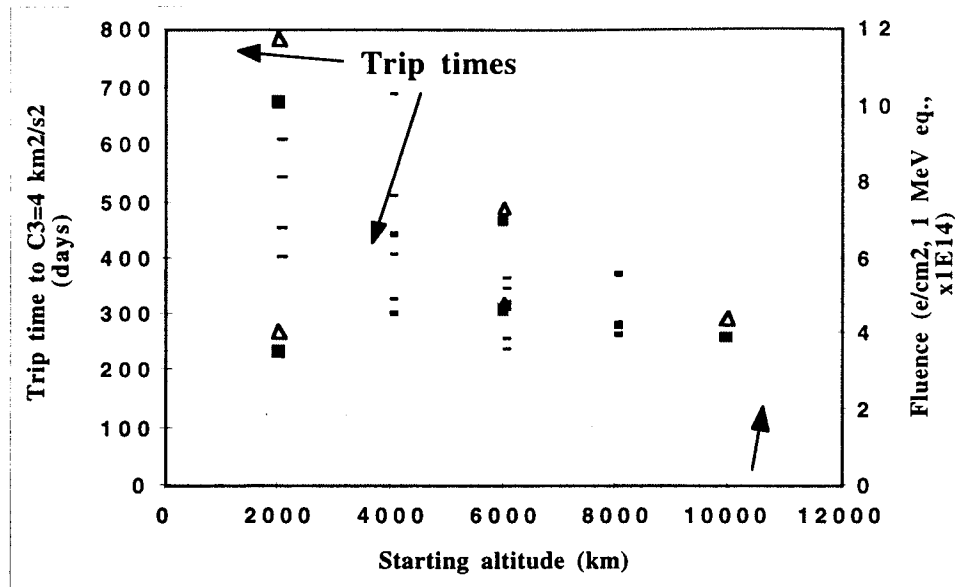


Figure 3. The envelope of trip time and starting altitude selected for radiation calculations.

The data shown in Figure 2 were used to estimate solar array degradation. The arrays were then sized to supply the desired power level at the end of the geocentric phase.

IV. Trajectories

IV.1 The Geocentric Phase

Geocentric trajectories were calculated to determine delivered mass as a function of trip time for each launch vehicle. Available launch vehicle data show only the mass delivery capability for each vehicle to a C3 of a little more than $-20 \text{ km}^2/\text{s}^2$, corresponding to trip times of less than a year. Therefore, the data were extrapolated down further to find performance capability out to 700 day trip times. While these data are likely accurate for the Delta-class launch vehicles, it was not determined whether the Atlas or Sea Launch vehicles could start from the required orbit for the longest trip times. In all

cases, initial orbits with a starting perigee altitude of 500 km were used. There is no trip-time benefit in starting from a circular parking orbit, though there may be a mass benefit (due to less radiation degradation of the solar arrays) if one started outside of the radiation belts. However, such a starting altitude would dictate a relatively large initial C3. Several circular orbit functions are shown for comparison, but were made using thruster performance parameters of 2000 s specific impulse and 270 mN of thrust for each engine.

It is clear from Figure 4 that Earth spiral trip times of about 1 year provide substantial payload delivery improvement over the mass injected directly to a C3 of $4 \text{ km}^2/\text{s}^2$ by each launch vehicle. However, a direct comparison of the mass deployed to Earth escape and the SEP curves is deceptive, as the latter includes the dry mass of the SEP module and its associated solar array.

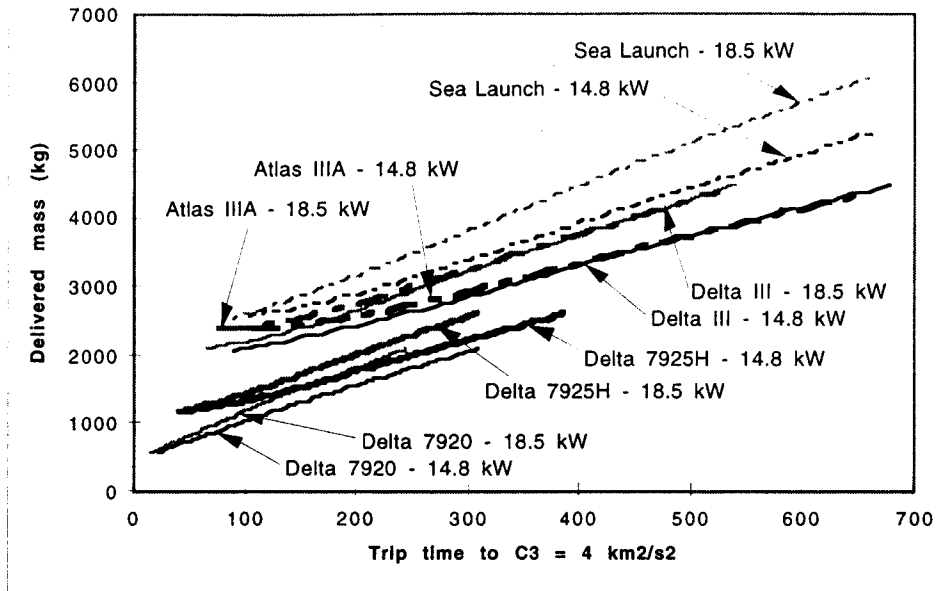


Figure 5. Delivered mass as a function of trip time for negative C3 launches and SEP.

The Heliocentric Phase: Europa as a Mission Example

Two trajectories were chosen; the first is an indirect Earth to Jupiter trajectory. The optimized delivered mass using this trajectory is a little over 2000 kg for an initial ($C3=4 \text{ km}^2/\text{s}^2$) mass of 2900. The trip time prior to executing a Jupiter-orbit-insertion maneuver is 4 years after Earth-escape. The second trajectory uses a single Venus gravity assist, and delivers an optimum mass of 2200 kg to Jupiter in 4 years. The initial mass for this trajectory is 3800 kg. No Earth-flyby trajectories were considered due to the probable use of radioisotope thermoelectric generators on the spacecraft.

IV.2.1 Europa Orbiter (2002-2004)

Though rapid trip times and the potential availability of a Shuttle IUS launch makes an entirely chemical propulsion approach to Europa Orbiter attractive, the possible availability of inexpensive SEP systems and lower cost launch vehicles provide an interesting alternative. The IUS delivers the Europa Orbiter spacecraft, currently in advanced planning stages, to a $C3$ of $81 \text{ km}^2/\text{s}^2$. The Velocity at Jupiter is then approximately 5.6 km/s, and a total Delta-V of about 2.6 km/s is needed for Jupiter orbit insertion, JPR, tour and navigation, and Europa orbit insertion. The current trajectory includes a Ganymede swing-by.

At present, the Europa Orbiter mission design calls for a spacecraft dry mass of approximately 400 kg delivered to Europa. Initial spacecraft wet mass integrated into the

IUS system would be approximately 980 kg - the IUS limit.

In order to equal this performance in terms of delivered mass, a negative $C3$ SEP scenario would have to deliver 1400 kg to Jupiter, and have an initial (launch) mass of 2100 kg. Trip time would be 1.5 years longer than the baseline mission.

IV.2.2 Europa Lander (2006-2009)

The Heliocentric trajectories were optimized for final mass for the cases shown in Table. Because of the significant increase in mass needed to perform a Europa lander mission, it is not possible to perform this mission using 2 or 4 NSTAR thrusters. Instead, trajectories were determined using a propulsion system equivalent to the use of 6 or 8 NSTAR ion engines, while assuming the use high power Hall thrusters; that is, the propulsion system component masses were accounted for as the same as for the 3.4 kW engines, (except, of course for the PPU and propellant tank). The trajectories are overly optimistic for existing technology, as most of the delivered mass would be composed of the SEP system. Also inherent in the assumption is equivalent system efficiency, as well as specific impulse and throttling profile.

Several approaches to a Europa lander spacecraft have been evaluated [ref]. For large (more than a few kilograms) science payload packages on the Jovian moon's surface, it was found that there is no advantage

to leaving a s/c in orbit around Europa for the lander mission. A mass of 660 kg would be needed in orbit around Europa to meet mission objectives - a challenge for a chemical propulsion system. This translates into a need for 1541 kg of mass delivered to Jupiter after discarding a SEP system and solar array.

In order to deliver more mass to Jupiter from a C3 of 4 km²/s², solar array power would have to be further increased, and either Earth gravity assist or Venus-Earth

combination gravity assist trajectories would have to be allowed. Either of these options would increase trip time and the Vinfinity at Jupiter, as was seen in the double Venus gravity assist example. Furthermore, this study is a testimonial for the need for high-power electric thrusters.

Jupiter Rendezvous Trajectories

Trajectory type	Departur e date	Flight time (years)	Initial mass @ C3 = 4 km ² /s ² (kg)	Delivered mass (kg)	Eq. ΔV (km/s)	Vhyp (km/s)
Indirect, 2 oper. thrust., 12-kW	Dec-04	5.3	1349	945	10.9	3.96
EGA, 2 oper. thrust., 12-kW	Jul-04	4.3	1908	1559	6.3	5.56
VEGA, 2 oper. thrust., 12-kW	Mar-04	4.6	1771	1528	4.8	5.84
VVGA, 2 oper. thrust., 12-kW	Mar-04	4.3	1694	1396	6.2	6.67
Indirect, 13.5-kW (4 NSTAR Eq.)	Jan-06	4.8	2020	1423	10.7	4.92
Indirect, 14.5-kW(6 NSTAR Eq.)	Jan-06	4.8	2315	1621	11.0	5.66
Indirect, 18.5-kW (6 NSTAR Eq.)	Jan-06	3.2	2880	2027	10.93	5.06
Indirect, 18.5-kW (8 NSTAR Eq.)	Jan-06	3.2	2989	2087	11.09	5.59
Indirect, ATLAS IIIA C3=11 km ² /s ² , 2 and 4 oper. thrust., 13.5-kW	Mar-06	4.8	2105	1677	7.2	5.22
VGA, 13.5-kW(4 NSTAR Eq.)	Sept-08	3.6	1817	1233	12.2	6.45
VGA, 18.5-kW(6 NSTAR Eq.)	Aug-08	2.7	2690	1825	12.35	6.44
VGA, 18.5-kW(8 NSTAR Eq.)	Aug-08	2.7	3240	2204	12.26	6.56
VGA, ATLAS IIIA C3=16.2 km ² /s ² , 2 and 4 oper. thrust., 13.5-kW	Oct-08	3.5	1881	1396	9.4	6.57

Table 8. Trajectories for the heliocentric mission phase.

Recommendations for Future Work

There are several refinements that could made to enhance the usefulness of the charts derived for this study. First, the geocentric trajectories could be run with a decreasing power level as a function of trip time. Here, only constant end-of-life solar array power was assumed during the Earth spiral trajectories. However, the beginning-of-life power for the 18.5 and 14.8 kW arrays is 23.1 and 18.5 kW, respectively. The additional power would reduce the trip time needed to deliver a desired mass to Earth escape.

The high power heliocentric trajectories are overly optimistic for state-of-the-art ion thruster flight hardware, and should be reevaluated using performance data from test thrusters currently under development by multiple organizations in the U.S. and abroad. In particular, experimental evaluation of Hall thruster lifetime at high (~3000 s) specific impulse should be undertaken.

The radiation trajectories should be run for the geotransfer orbits instead of the circular orbits used in this study. Doing so would reduce the charged particle fluence used to calculate solar array degradation. Conversely, the trajectory codes used for the heliocentric trajectory calculations should be modified to allow for degradation of performance over the trip time.

Experimentally, an effort to take thrusters developed for Earth orbital applications and modify them for higher specific impulse operation without sacrificing needed lifetime may prove beneficial for NASA mission applications.

Conclusions

Regardless of the final destination, this study provides a tool for comparing several launch vehicles that may be used to deliver spacecraft to a negative C3; given a required spacecraft mass, the spiral time necessary to deliver that mass to Earth escape using a SEP system can be determined.

The use of negative C3 launches and SEP may allow delivery of large spacecraft to deep space destinations using less expensive launch vehicles than would otherwise be required. This study assumes the availability of large solar arrays and high specific impulse Hall thrusters for deep space operation.

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